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Linder et al.

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[54] SPACECRAFT MOMENTUM UNLOADING USING CONTROLLED MAGNETIC TORQUES

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[73] Assignee: General Electric Company, East Windsor, N.J.

[21] Appl. No.: 488,919

[22] Filed: Mar. 5, 1990

[51] Int. Cl.[5] B64G 1/10

[52] U.S. Cl. 244/166

[58] Field of Search 244/165, 166, 164, 169, 244/170, 171, 172; 364/434, 459

[56] References Cited

U.S. PATENT DOCUMENTS

3,429,524	2/1969	Buckingham et al.	244/1
3,681,583	8/1972	Kubo et al.	244/166
3,834,653	9/1974	Perkel	244/166
3,955,858	5/1976	Poubeau	244/166
4,010,921	3/1977	Pistiner et al.	244/166
4,062,509	12/1977	Muhlfelder et al.	244/166
4,071,211	1/1978	Muhlfelder et al.	244/165
4,916,622	6/1988	Raman et al.	244/166

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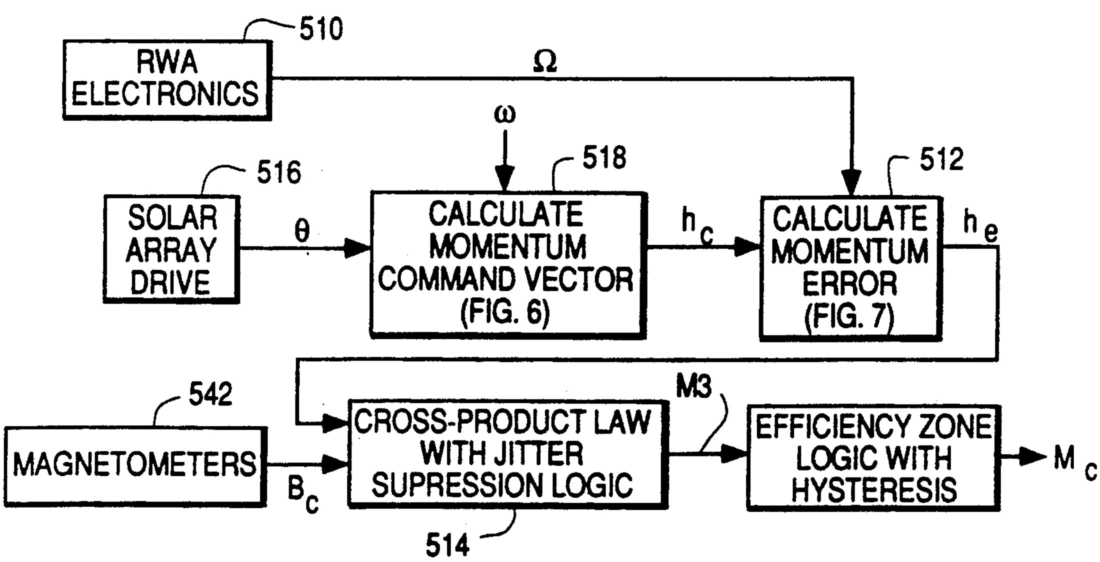
Assistant Examiner—Stephen P. Avila
Attorney, Agent, or Firm—William H. Meise

[57] ABSTRACT

A method for maintaining the attitude of a three-axis controlled satellite by use of magnetic torquers includes using magnetometers for measuring the direction of the ambient geomagnetic field. The direction of the net reaction wheel momentum is also determined. The angle between the direction of the geomagnetic field and the net reaction wheel momentum is determined. The angle is compared with a threshold value. Magnetic torquer power consumption is reduced by operating the magnetic torquers only when the angle exceeds the threshold value.

5 Claims, 7 Drawing Sheets



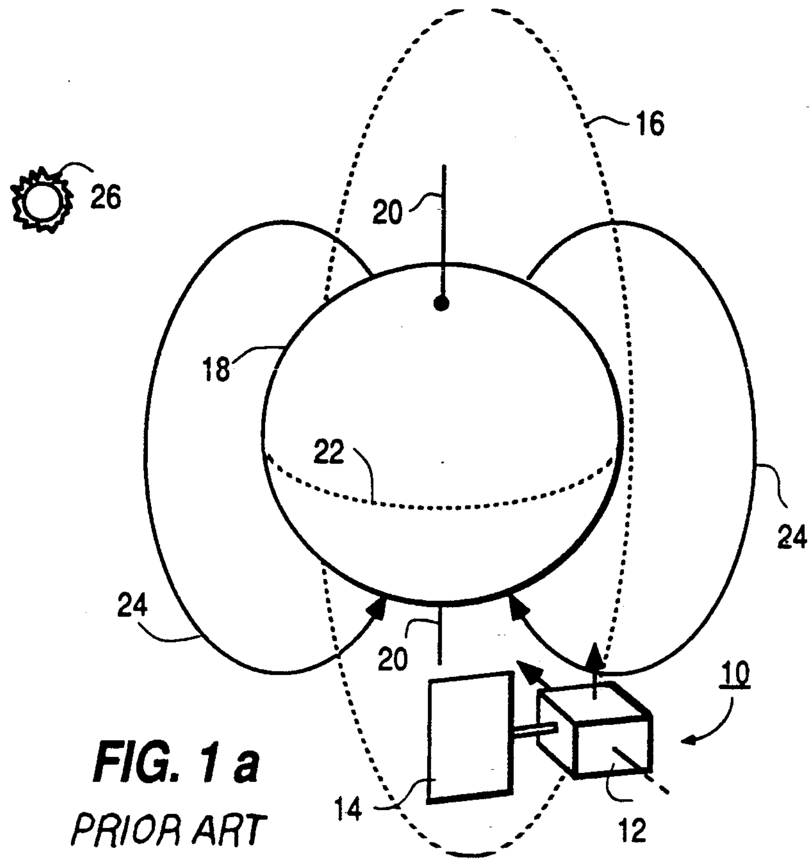
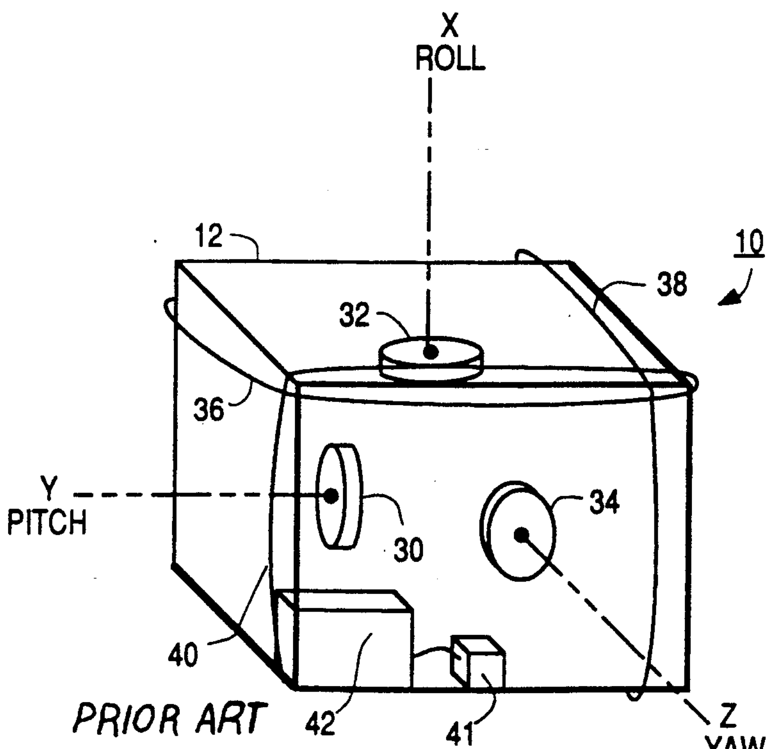


FIG. 1 a
PRIOR ART



PRIOR ART
FIG. 1b

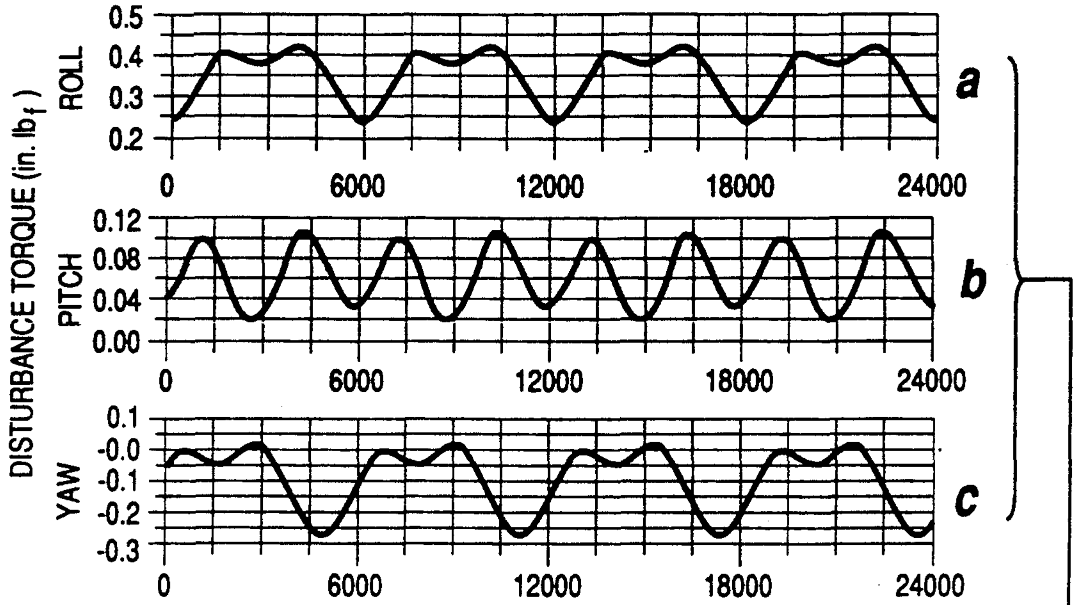


FIG. 2
PRIOR ART

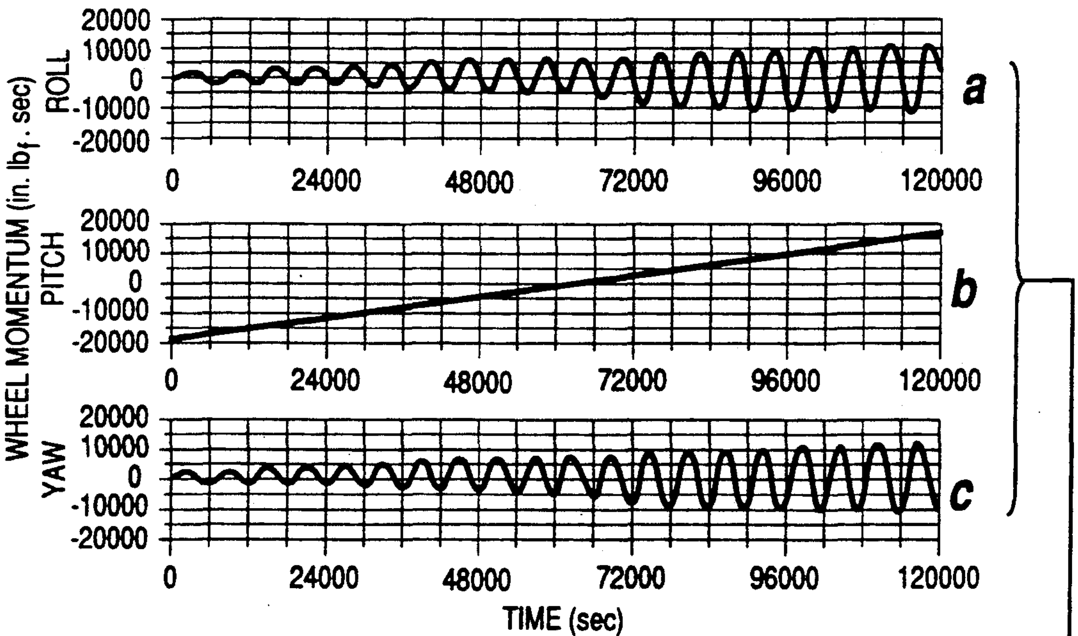


FIG. 3
PRIOR ART

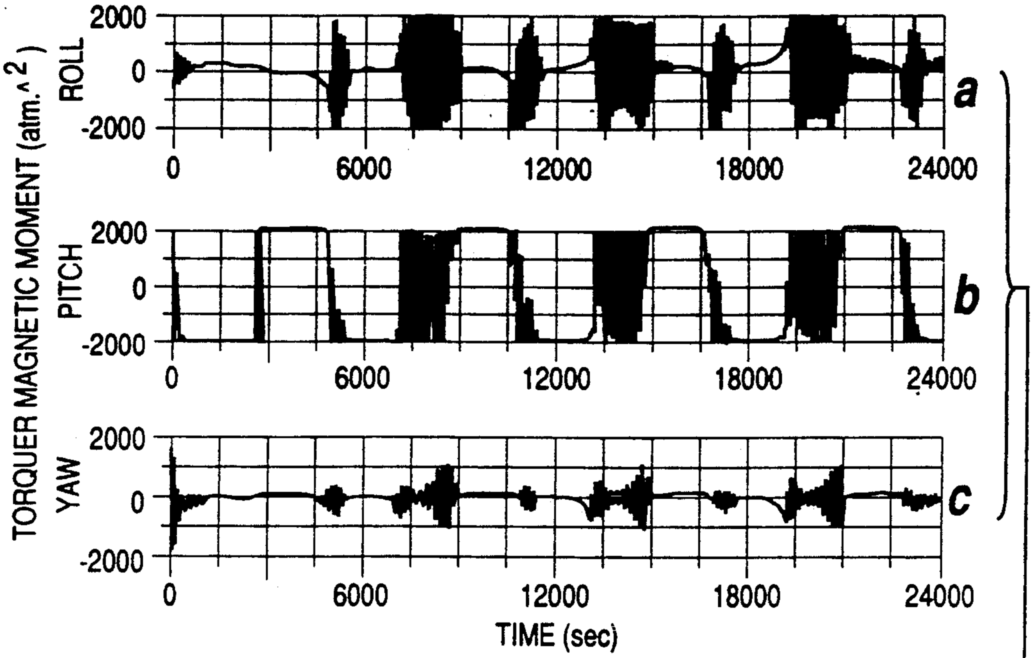


FIG. 4
PRIOR ART

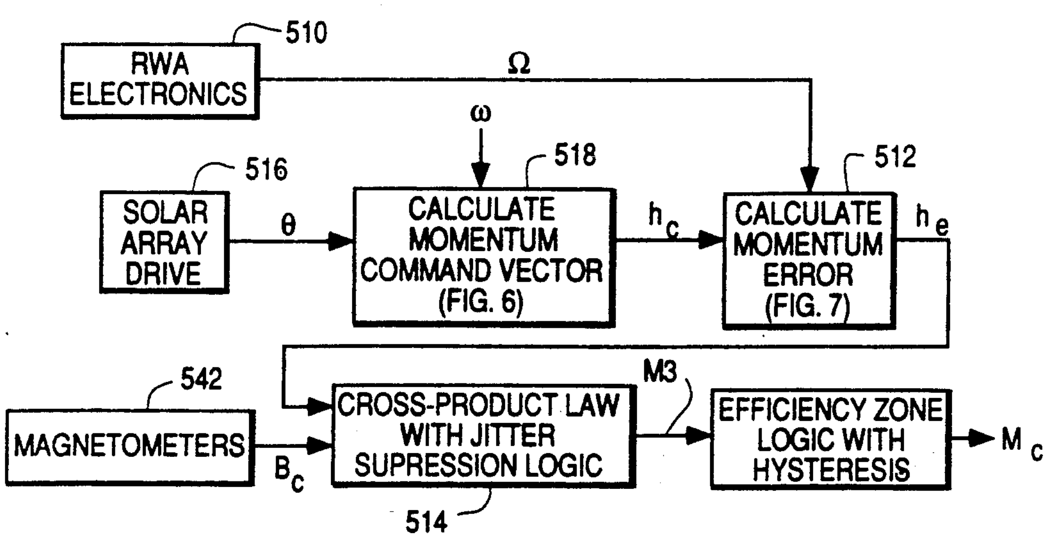


FIG. 5

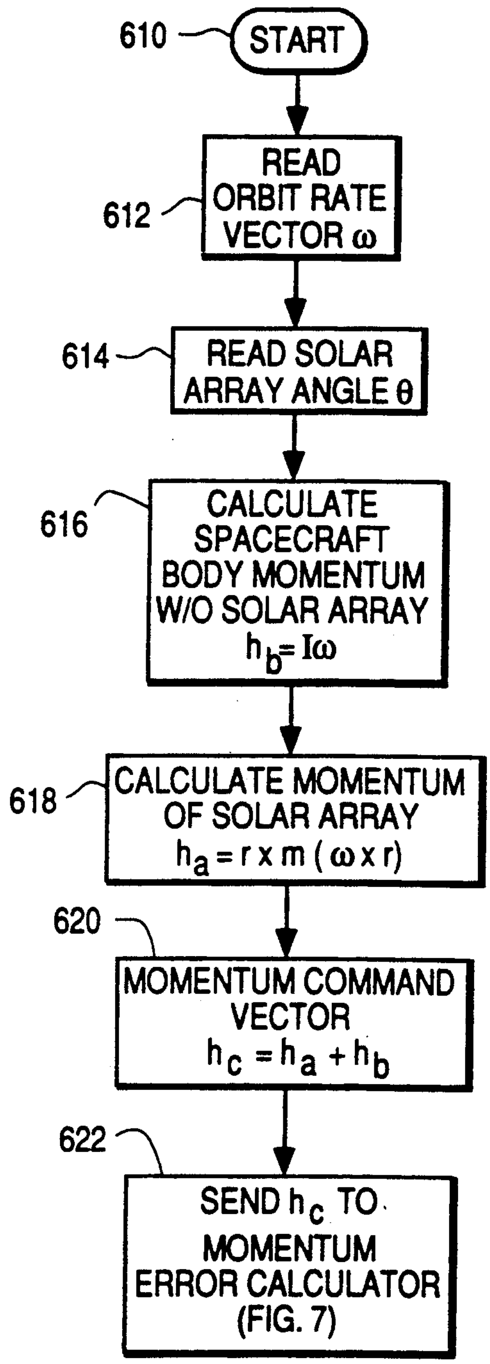

610
START
612
READ
ORBIT RATE
VECTOR ω
614
READ SOLAR
ARRAY ANGLE θ
616
CALCULATE
SPACECRAFT
BODY MOMENTUM
W/O SOLAR ARRAY
$h_b = I\omega$
618
CALCULATE MOMENTUM
OF SOLAR ARRAY
$h_a = r \times m(\omega \times r)$
620
MOMENTUM COMMAND
VECTOR
$h_c = h_a + h_b$
622
SEND h_c TO
MOMENTUM
ERROR CALCULATOR
(FIG. 7)

FIG. 6

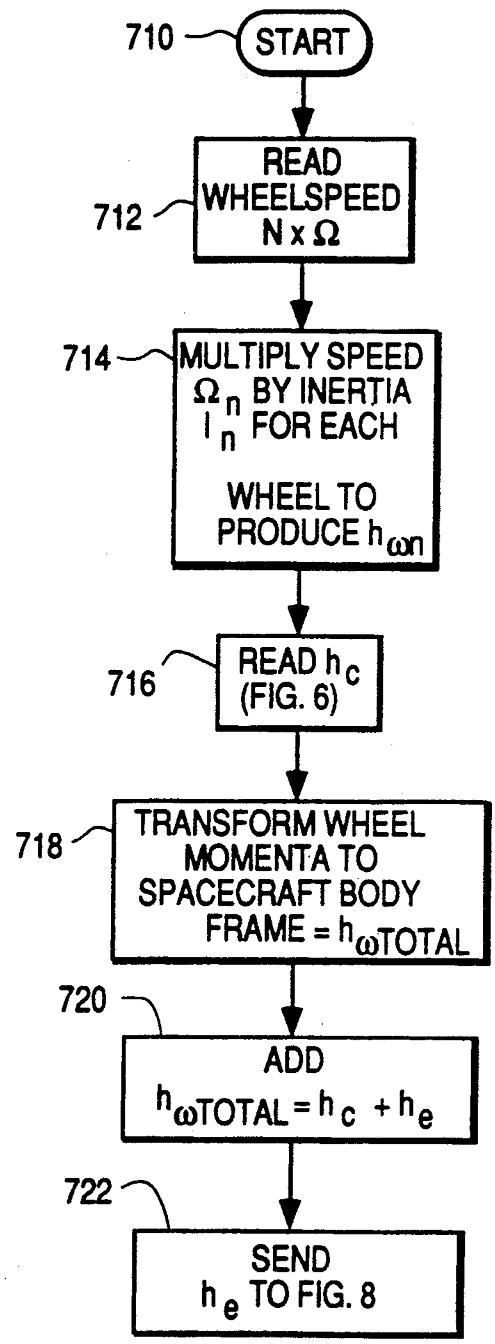
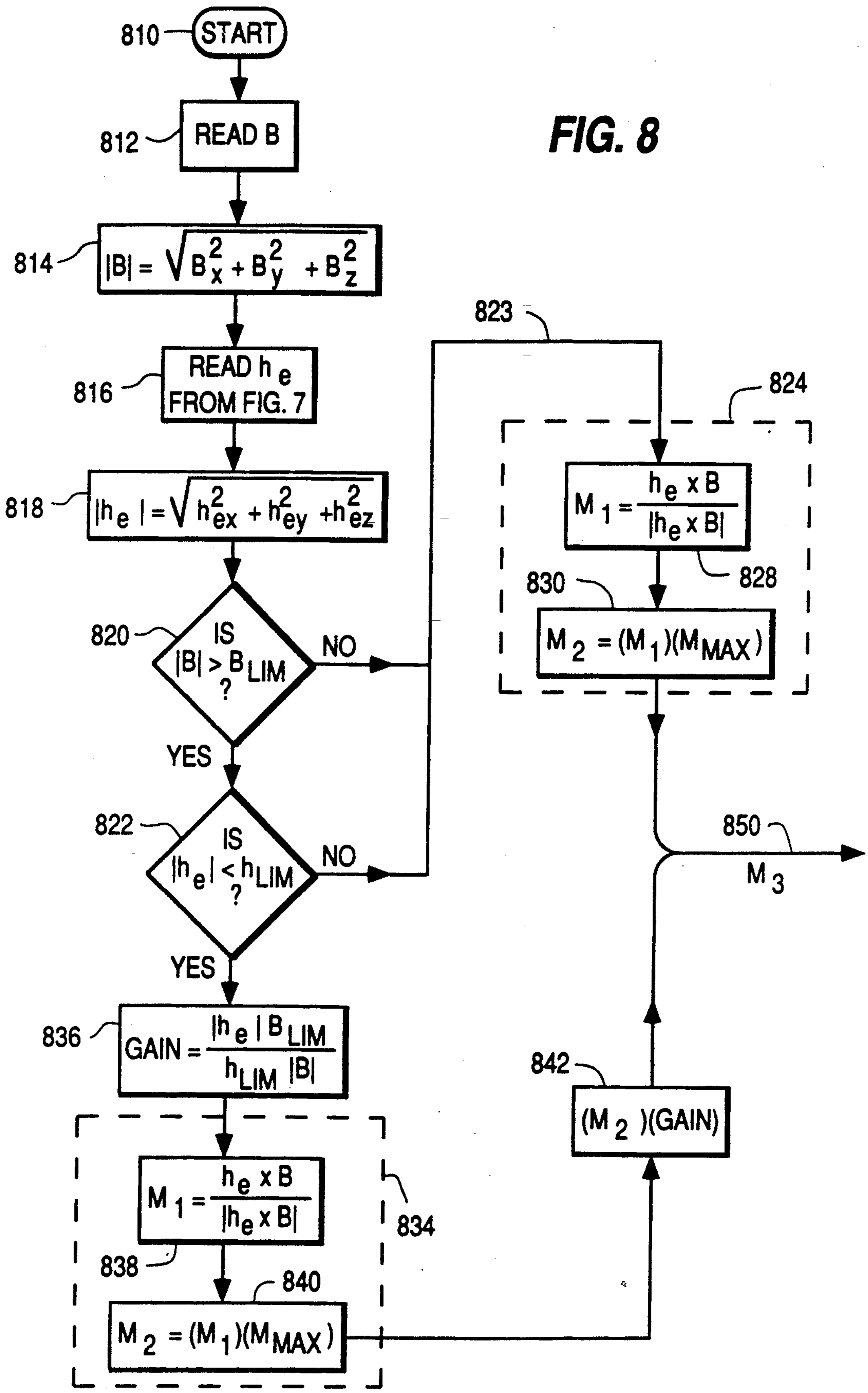

710
START
712
READ
WHEELSPEED
$N \times \Omega$
714
MULTIPLY SPEED
Ω_n BY INERTIA
I_n FOR EACH
WHEEL TO
PRODUCE $h_{\omega n}$
716
READ h_c
(FIG. 6)
718
TRANSFORM WHEEL
MOMENTA TO
SPACECRAFT BODY
FRAME = $h_{\omega TOTAL}$
720
ADD
$h_{\omega TOTAL} = h_c + h_e$
722
SEND
h_e TO FIG. 8

FIG. 7



810
START
812
READ B
814
$|B| = \sqrt{B_x^2 + B_y^2 + B_z^2}$
816
READ h_e FROM FIG. 7
818
$|h_e| = \sqrt{h_{ex}^2 + h_{ey}^2 + h_{ez}^2}$
820
IS $|B| > B_{LIM}$?
NO
YES
822
IS $|h_e| < h_{LIM}$?
NO
YES
836
$GAIN = \frac{|h_e| B_{LIM}}{h_{LIM} |B|}$
834
$M_1 = \frac{h_e \times B}{|h_e \times B|}$
838
840
$M_2 = (M_1)(M_{MAX})$
823
824
$M_1 = \frac{h_e \times B}{|h_e \times B|}$
828
830
$M_2 = (M_1)(M_{MAX})$
850
M_3
842
$(M_2)(GAIN)$

FIG. 8

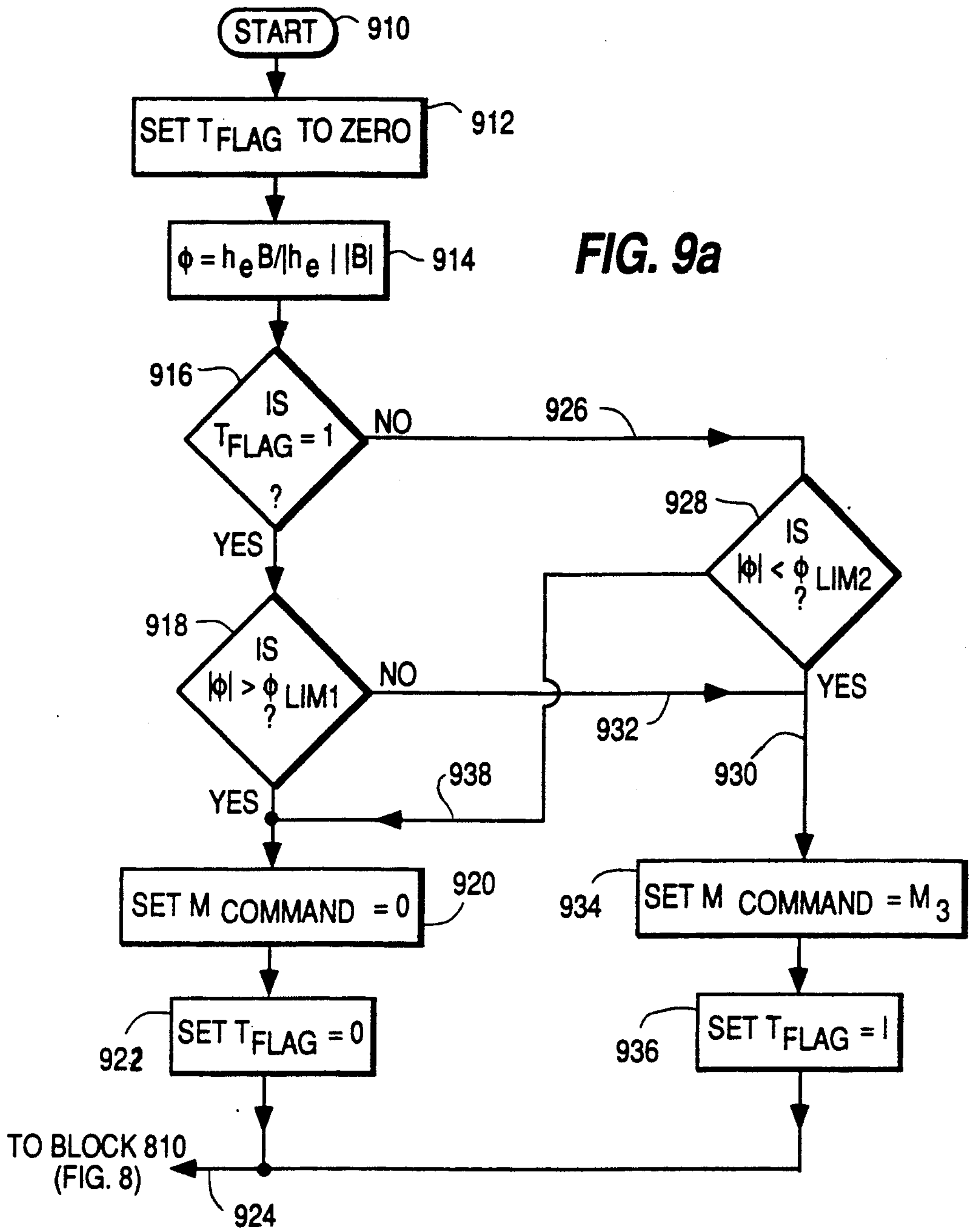

START
910
SET T_{FLAG} TO ZERO
912
$\phi = h_e B/|h_e||B|$
914
916
IS $T_{FLAG} = 1$?
NO
926
YES
928
IS $|\phi| < \phi_{LIM2}$?
918
IS $|\phi| > \phi_{LIM1}$?
NO
YES
932
930
938
YES
SET M COMMAND = 0
920
934
SET M COMMAND = M_3
SET T_{FLAG} = 0
922
936
SET T_{FLAG} = I
TO BLOCK 810 (FIG. 8)
924

FIG. 9a

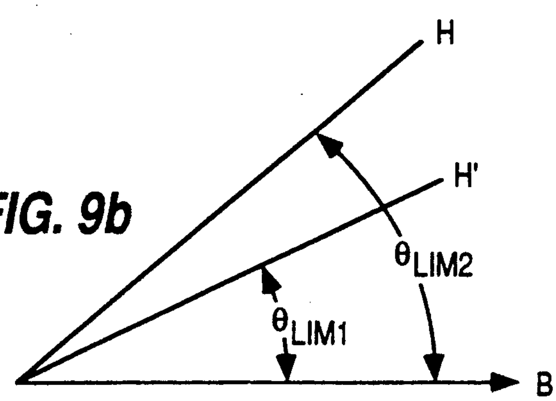
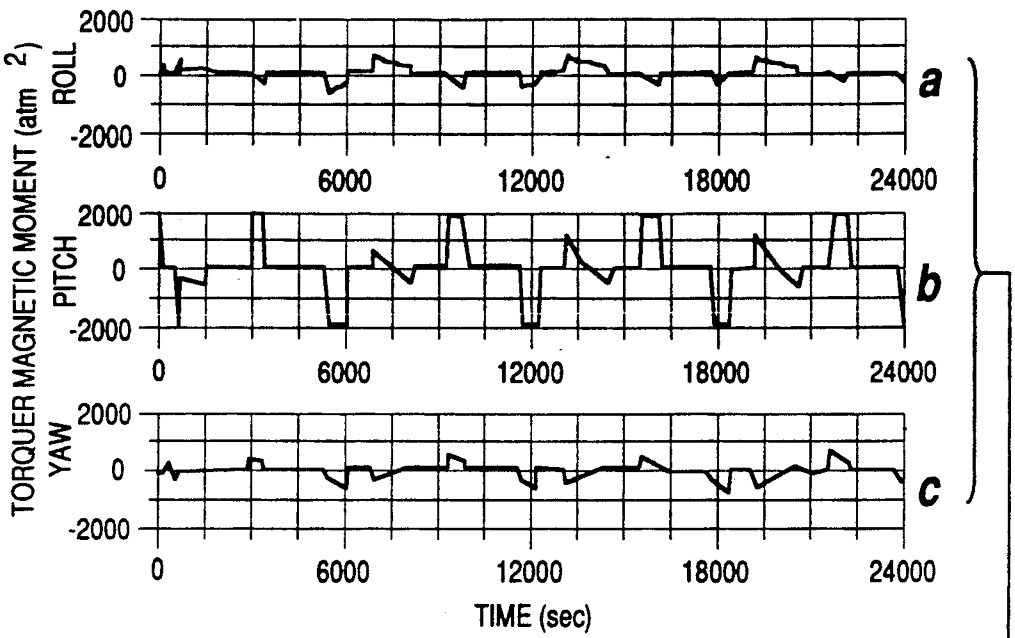
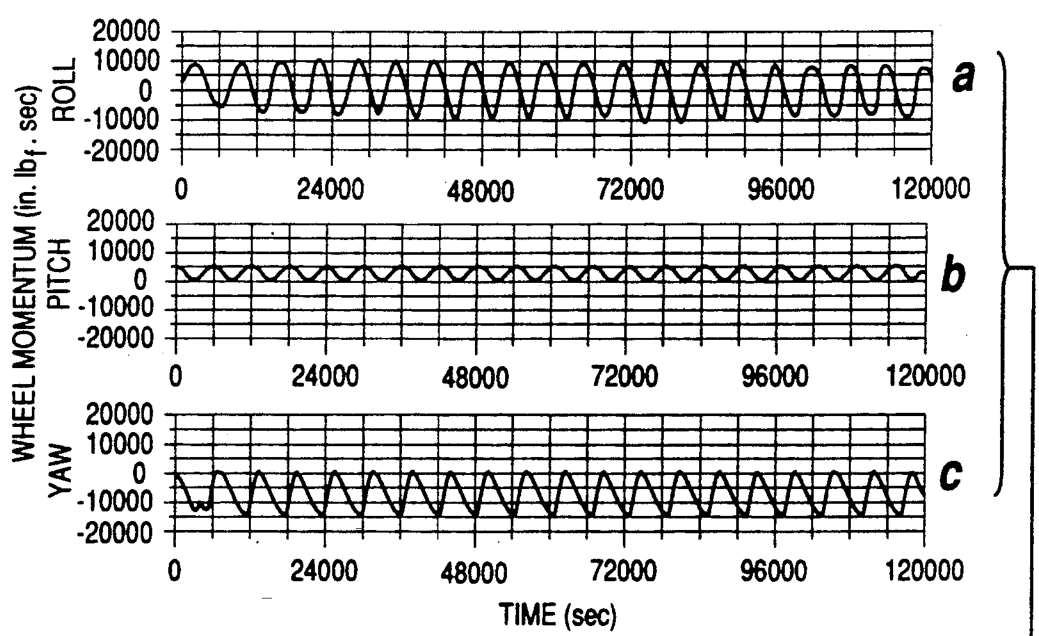

H
H'
θ_{LIM2}
θ_{LIM1}
B

FIG. 9b



TORQUER MAGNETIC MOMENT (atm 2)
ROLL
PITCH
YAW
2000
0
-2000
0
6000
12000
18000
24000
TIME (sec)
a
b
c

FIG. 10



WHEEL MOMENTUM (in. lbf . sec)
ROLL
PITCH
YAW
20000
10000
0
-10000
-20000
0
24000
48000
72000
96000
120000
TIME (sec)
a
b
c

FIG. 11

SPACECRAFT MOMENTUM UNLOADING USING CONTROLLED MAGNETIC TORQUES

The invention described herein was made in the performance of work under NASA Contract No. NAS5-32000 and is subject to the provisions of Section 305 of the National Aeronautics and Space Act of 1958, as amended (72 STAT. 435; 42 U.S.C. 2457).

BACKGROUND OF THE INVENTION

This invention relates to attitude control of spacecraft, and more particularly to unloading of reaction or momentum wheel momentum by the use of controlled magnetic torquers.

Orbiting spacecraft are used for a large variety of sensing and communication purposes for photographic purposes, it is desirable for the spacecraft to be relatively near the Earth, so that the cameras or sensors are close to the subjects, and so that the spacecraft may cover large amounts of surface by virtue of a rapid orbital rate. For communication purposes, a geosynchronous equatorial orbit is often desirable. Whatever the orbit, the satellite must be stabilized in space if the sensors or antennas are to be pointed in appropriate direction.

Spacecraft attitude stabilization may be accomplished by spinning the spacecraft and by mounting the sensors or antennas on a despun platform. Alternatively, the spacecraft may be stabilized in three axes. Three-axis stabilization may be accomplished by a control system using fuel-burning thrusters, but the use of such thrusters requires the expenditure of fuel, which tends to limit the service life of the spacecraft. Another method of three-axis stabilization uses reaction wheels or momentum wheels mounted within the spacecraft. The reaction wheels are torqued under the control of an attitude control system to cause the spacecraft body to assume the desired attitude. The energy for torquing the wheels is electrical, and is derived from solar panels. Attitude control by the use of wheels therefore generally provides longer operating lifetime than simple attitude control by the use of thrusters.

In order to maintain a face of the satellite directed toward the Earth, the satellite must rotate about its pitch axis at the orbit rate. When using reaction wheel control, the wheel or wheels are torqued to maintain an earth-oriented attitude. Environmental disturbance torques that act on the spacecraft tend to result in a gradual increase in the wheel momenta. In order to prevent the wheels from reaching their mechanical momentum limits, and to thereby maintain the ability to provide reaction wheel based attitude control, the wheel momentum must be periodically reduced. Wheel momentum may be reduced by the use of fuel-consuming thrusters which torque the spacecraft body in a manner which opposes the wheel momentum. However, the use of such thrusters is subject to the same objection of limited lifetime described above.

FIG. **1***a* illustrates a spacecraft designated generally as **10**, and including a body **12** and a solar panel **14**. Spacecraft **10** is following an orbit illustrated by dashed-line path **16** about the Earth **18**. The Earth's rotational axis is illustrated as **20**, and the Equator is illustrated by dash-line **22**. Lines of magnetic force associated with the Earth's magnetic field are illustrated as **24**. As illustrated in FIG. **1**, solar panel **14** is directed toward the sun **26**, and is controlled to continue facing the sun at all orbital positions of satellite **10**.

FIG. **1***b* illustrates details of spacecraft **10**. In FIG. **1***b* a first reaction wheel **30** is oriented with its axis along the pitch or y axis, a second reaction wheel **32** is oriented with its axis along the x or roll axis, and a third reaction wheel **34** is oriented with its axis along the yaw or z axis. Also in FIG. **1***b*, magnetic torquer windings illustrated as **36**, **38** and **40** are wound about the spacecraft body. Further in FIG. **1***b*, a set of three-axis magnetometers is illustrated as a block **41**, and is connected to a torquer control arrangement illustrated as a block **42**

FIGS. **2***a*, **2***b* and **2***c* illustrate as time functions the roll, pitch and yaw disturbance torques, respectively, measured in inch-lb. In FIG. **2**, the disturbance torques include a periodic component and an offset component. The disturbance torques arise from atmospheric drag, gravity gradient effects, and solar pressure. The cyclical perturbations recur at roughly 6,000 second intervals, corresponding to the orbital period. The constant offset of the disturbance torques is attributable to the asymmetric orientation of solar panel **14** of FIG. **1***a*. In FIG. **2**, zero seconds and each 6,000 second increment thereafter represent the times at which the spacecraft crosses the Earth's Equator, known as the ascending node.

FIGS. **3***a*, **3***b* and **3***c* represent as time functions the roll, pitch and yaw wheel momentum, respectively, in the absence of unloading. The changes in wheel momentum with time are attributable to the disturbance torques illustrated in FIG. **2**. Some components of the disturbance torques of FIG. **2** contribute to the cyclical variations of wheel momentum in FIG. **3**, while others result in secular or continually increasing or decreasing components. For example, in FIG. **3***a*, the peak values of the cyclical component of wheel momentum increases with increasing time. At some point, therefore, such a continuing increase in amplitude will result in reaching the wheel momentum limit. Similarly, the pitch wheel momentum of FIG. **3***b* increases monotonically with increasing time. In order to maintain continuous attitude control, the secular component of the wheel momentum must be unloaded.

FIGS. **4***a*, **4***b* and **4***c* represent the roll, pitch and yaw torquer magnetic moment measured in ampere-turns-meter^2 (ATM^2) for a spacecraft using a prior art torquer control system which pulse width or continuously modulates the torquer current. The prior art control system measures the Earth's geomagnetic field B and the net momentum of the reaction wheel, and forms a vector cross product HXB. The normalized HXB vector is multiplied by the maximum magnetic moment which can be achieved along each of the roll, pitch and yaw axes to determine the torquer dipole command. In FIGS. **4***a*, **4***b* and **4***c*, large excursions or jitter in the roll and pitch moments, and lesser excursions in the yaw moments, occur at about 7500 seconds, 14,000 seconds, and 20,000 seconds, corresponding to passage of the spacecraft across the Earth's magnetic poles. The jitter illustrated in FIG. **4** arises because the magnitude of the Earth's geomagnetic field near the poles is large, but the cyclical component of the wheel momentum results in relatively low wheel momentum. Operation of the magnetic torquers in regions near the poles, therefore, produces a high magnitude of wheel momentum change at times when the wheel momentum is low, which tends to drive the torquer command from one extreme to the other. This in turn results in the illustrated jitter, which

disturbs the spacecraft attitude. Such disturbances are undesirable.

SUMMARY OF THE INVENTION

A method for maintaining the attitude of the satellite includes the step of measuring the direction of the ambient magnetic field in the vicinity of the satellite. The magnitude and direction of the net wheel momentum of the satellite is also determined, and the angle therebetween is determined. The angle is compared with a predetermined value, and magnetic torquers are energized only when the angle exceeds a predetermined value. In a particular embodiment of the invention, the magnitude of the ambient magnetic field is also measured.

DESCRIPTION OF THE DRAWING

FIG. **1***a* is a perspective or isometric view of an asymmetric satellite in a polar orbit around the Earth, with its solar panel pointed toward the Sun, and FIG. **1***b* illustrates the orientation of reaction or momentum wheels and magnetic torquing coils in the spacecraft of FIG. **1***b*, FIGS. **1***a* and **1***b* are referred to jointly as FIG. **1**;

FIGS. **2***a*, **2***b* and **2***c*, referred to jointly as FIG. **2**, are plots versus time of environmental disturbance torques about roll, pitch and yaw axes, respectively;

FIGS. **3***a*, **3***b* and **3***c*, referred to jointly as FIG. **3**, are plots versus time of wheel momentum about roll, pitch and yaw axes, respectively, without momentum unloading control;

FIGS. **4***a*, **4***b* and **4***c*, referred to jointly as FIG. **4**, are plots versus time of magnetic moments produced about roll, pitch and yaw axes, respectively, by spacecraft torquers using a prior-art control system;

FIG. **5** is a simplified block diagram of a control system according to the invention;

FIGS. **6**, **7**, **8** and **9***a* are simplified flow charts illustrating details of the control system of FIG. **5**, and FIG. **9***b* illustrates various angles associated with the flow chart of FIG. **9***a;*

FIGS. **10***a*, **10***b* and **10***c*, referred to jointly as FIG. **10**, are plots versus time of magnetic moments produced about roll, pitch and yaw axes, respectively, by spacecraft torquers using a control system according to the invention;

FIGS. **11***a*, **11***b* and **11***c*, referred to jointly as FIG. **11**, are plots versus time of wheel momentum about roll, pitch and yaw axes, respectively, using momentum control according to the invention.

DESCRIPTION OF THE INVENTION

FIG. **1***a* illustrates a spacecraft designated generally as **10**, and including a body **12** and a solar panel **14**. Spacecraft **10** is following an orbit illustrated by dashed-line path **16** about the Earth **18**. The Earth's rotational axis is illustrated as **20**, and the Equator is illustrated by dash-line **22**. Lines of magnetic force associated with the Earth's magnetic field are illustrated as **24**. As illustrated in FIG. **1**, solar panel **14** is directed toward the sun **26**, and is controlled to continue facing the sun at all orbital positions of satellite **10**.

FIG. **1***b* illustrates details of spacecraft **10**. In FIG. **1***b* a first reaction wheel **30** is oriented with its axis along the pitch or y axis, a second reaction wheel **32** is oriented with its axis along the x or roll axis, and a third reaction wheel **34** is oriented with its axis along the yaw or z axis. Also in FIG. **1***b*, magnetic torquer windings illustrated as **36**, **38** and **40** are wound about the spacecraft body. Further in FIG. **1***b*, a set of three-axis magnetometers is illustrated as a block **41**, and is connected to a torquer control arrangement illustrated as a block **42**

FIGS. **2***a*, **2***b* and **2***c* illustrate as time functions the roll, pitch and yaw disturbance torques, respectively, measured in inch-lb. In FIG. **2**, the disturbance torques include a periodic component and an offset component. The disturbance torques arise from atmospheric drag, gravity gradient effects, and solar pressure. The cyclical perturbations recur at roughly 6,000 second intervals, corresponding to the orbital period. The constant offset of the disturbance torques is attributable to the asymmetric orientation of solar panel **14** of FIG. **1***a*. In FIG. **2**, zero seconds and each 6,000 second increment thereafter represent the times at which the spacecraft crosses the Earth's Equator, known as the ascending node.

FIGS. **3***a*, **3***b* and **3***c* represent as time functions the roll, pitch and yaw wheel momentum, respectively, in the absence of unloading. The changes in wheel momentum with time are attributable to the disturbance torques illustrated in FIG. **2**. Some components of the disturbance torques of FIG. **2** contribute to the cyclical variations of wheel momentum in FIG. **3**, while others result in secular or continually increasing or decreasing components. For example, in FIG. **3***a*, the peak values of the cyclical component of wheel momentum increases with increasing time. At some point, therefore, such a continuing increase in amplitude will result in reaching the wheel momentum limit. Similarly, the pitch wheel momentum of FIG. **3***b* increases monotonically with increasing time. In order to maintain continuous attitude control, the secular component of the wheel momentum must be unloaded.

FIGS. **4***a*, **4***b* and **4***c* represent the roll, pitch and yaw torquer magnetic moment measured in ampere-turns-meter^2 (ATM^2) for a spacecraft using a prior art torquer control system which pulse width or continuously modulates the torquer current. The prior art control system measures the Earth's geomagnetic field B and the net momentum of the reaction wheel, and forms a vector cross product HXB. The normalized HXB vector is multiplied by the maximum magnetic moment which can be achieved along each of the roll, pitch and yaw axes to determine the torquer dipole command. In FIGS. **4***a*, **4***b* and **4***c*, large excursions or jitter in the roll and pitch moments, and lesser excursions in the yaw moments, occur at about 7500 seconds, 14,000 seconds, and 20,000 seconds, corresponding to passage of the spacecraft across the Earth's magnetic poles.

FIG. **5** is a simplified block diagram of a control system according to the invention. In FIG. **5**, reaction wheel electronics **510** measure reaction wheel speeds Ω. The Ω signals are coupled to a block **512**. A solar array drive is illustrated as a block **516**, and the angle θ of the solar array is coupled to a further block **518** which represents calculation of a momentum command vector h_c. The momentum command vector is applied to block **512**, together with wheel speeds Ω. Block **512** calculates momentum error h_e. The momentum error h_e calculated in block **512** is applied to an input of a block **514**. A set of magnetometers for measuring three-axis geomagnetic field B is illustrated as a block **542**. Block **542** is coupled to a further input of block **514**.

Block **514** represents jitter suppression and calculation of a cross-product to produce a gain-controlled torquer magnetic moment command. The command is

applied to a further block **520**, which represents efficiency zone and hysteresis logic.

FIG. **6** is a simplified flow chart illustrating the calculation of momentum command vector h_c in block **518** of FIG. **5**. In FIG. **6**, the logic flow begins at a START block **610** and continues to a block **612**, which represents reading of the orbit rate vector ω. The orbit rate is known from orbit dimensions, and may be preprogrammed into a satellite or provided by an uplink from an Earth station. Block **614** represents the reading of the solar array angle θ. Block **616** represents calculation of the spacecraft body momentum h_b, which is the momentum of the spacecraft body, excluding the solar array. The calculation is performed as a product, where I represents the inertia matrix of the spacecraft body

$$h_b = I\omega \tag{1}$$

Block **618** represents the calculation of the momentum h_a of the solar array according to the equation

$$h_a = rXm((\omega Xr) + \dot{r}) \tag{2}$$

where

r is the position vector from the spacecraft center of mass to the solar array center of mass;

x is the vector cross product;

m is the mass of the solar array; and

$\dot{r}$ is the time derivative of $\dot{r}$, and both r and r may be functions of θ.

Block **620** represents the calculation of momentum command vector h_c by summing

$$h_c = h_a + h_b \tag{3}$$

Finally, block **622** represents the storage of momentum command vector h_c in order to make it available for further calculation.

FIG. **7** represents the calculation of momentum error h_e in block **512** of FIG. **5**. In FIG. **7**, the logic begins with a START block **710**, and proceeds to a block **712** which reads the value of Ω for each of the N wheels of the reaction wheel assembly. The logic proceeds to a block **714**, which represents the calculation of momentum for each of the wheels by multiplying the speed ω_n of each wheel by the inertia I_n for the corresponding wheel, to produce $h_{\omega n}$

$$h_{\omega n} = I_n \omega_n \tag{4}$$

Block **716** represents the fetching or reading of momentum command vector h_c from block **622** of FIG. **6**. Block **718** represents the transformation of the momenta of the n reaction wheels to the spacecraft body frame. The transformation matrix is based upon the geometry of the various wheel axes, and may be stored as data in the spacecraft or transmitted to the spacecraft from an earth station. The transformed wheel momenta h_{wTOTAL} is added to the momentum command vector h_c to produce the desired momentum error h_e in block **720**:

$$h_e = h_c + h_{wTOTAL} \tag{5}$$

Block **722** represents the storage of the calculated Value of h_e to make it available for further calculations.

FIG. **8** is a simplified block diagram representing the calculations performed in block **514** of FIG. **5**. In FIG. **8**, the logic begins at a START block **810**, and proceeds to a block **812**, which represents the reading of B, which is a vector representing the local geomagnetic fields. The logic proceeds to a block **814**, which represents the calculation of the magnitude of vector B by taking the square root of the sum of the squares of the components along the X, Y and Z axes

$$|B| = \sqrt{B_x^2 + B_y^2 + B_z^2} \tag{6}$$

Block **816** represents the reading of momentum error h_e from FIG. **7**. Block **818** represents the calculation of $|h_e|$ by taking the square root of the sum of the squares of the components along the X, Y and Z axes

$$|h_e| = \sqrt{h_{ex}^2 + h_{ey}^2 + h_{ez}^2} \tag{7}$$

The logic then arrives at a decision block **820**, in Which the magnitude of B from block **814** is compared with a limit value of B_{lim}. If the absolute magnitude of the geomagnetic fields $|B|$ is less than the limit value B_{lim}, the logic exits block **820** by the NO path and proceeds to a plurality of logic blocks illustrated together as **824**. If $|B|$ is greater than B_{lim}, the logic flows from decision block **820** by the YES block to a further decision block **822**. Block **822** represents the comparison of $|h_e|$ with a limiting value h_{lim}. If $|h_e|$ is greater than h_{lim}, the logic leaves decision block **822** by the NO path and flows to the logic represented by block **824**. Thus, so long as the absolute value of the geomagnetic field is less than the limiting value, which might be the case far from the Earth's magnetic poles, or if the absolute value of momentum error h_e is greater than a limiting value, the logic flows by path **823** to the logic within block **824**. Within block **824** the logic flows to a further block **828** which represents calculation of an intermediate momentum value M_1 as the quotient of vector cross products

$$M_1 = h_e XB/|h_e XB| \tag{8}$$

Logic block **830** calculates a further momentum product designated M_2

$$M_2 = (M_1)(M_{max}) \tag{9}$$

and this value of momentum is made available as M_3 on a logic path **850** for further calculations.

As mentioned, the magnitude of the geomagnetic or B fields near the poles is liable to be large while the magnitude of the net momentum is small. Under this condition, the logic flow through decision blocks **820** and **822** of FIG. **8** will result in a logic flow from the YES output of decision block **822** to a block **836**. Block **836** represents the calculation of a system gain according to the Equation

$$Gain = |h_e| B_{lim}/h_{lim}|B| \tag{10}$$

The logic flows from block **836** to further logic contained within dotted lines **834**. Logic blocks **838** and **840** contained within logic block **834** are identical to logic blocks **828** and **830** contained within logic block **824**. Product M_2 produced at the output of logic block **840** is applied to a further logic block **842**, which multiplies M_2 by the gain calculated in block **836**. The output of logic block **842** is coupled as M_3 onto logic path **850**. The values of M_3 appearing on logic path **850** are alter-

natives which depend upon conditions. If the magnitude of the environmental magnetic field is small, or if the momentum error is large, M_3 has a value established by the processing of block **824**, whereas if the momentum error is small and the magnetic field is large, the value is additionally multiplied by a gain function which is calculated in block **836**.

As so far described, the control system measures the magnitudes of the environmental magnetic field and the magnitude of the momentum error h_e for controlling system gain. This prevents the jitter illustrated in FIG. **4**, and provides improved pointing accuracy.

FIG. **9** is a simplified flow chart of processing according to the invention. The arrangement of FIG. **9***a* implements a hysteresis control which eliminates rapid cycling at the transition between torquing and non-torquing regions. Referring to FIG. **9***b*, a momentum direction H' makes an angle of θ_{LIM1} with the measured environmental magnetic field B. The magnitude of angle θ_{LIM1} is selected so that the magnetic torquing is sufficiently effective or efficient. A suitable angle might be 45 degrees. If torquing were simply proscribed for angles less than θ_{LIM1} and permitted for angles greater than or equal to θ_{LIM1}, noise and/or measurement errors might cause repeated ON-OFF cycling of the torquers when the angle between h_e and B is near θ_{LIM1}. This cycling is avoided by including a "dead zone" in the control. In FIG. **9***b*, the "dead zone" is the region between θ_{LIM1} and θ_{LIM2}. The torquers are not actuated when the angle lies between θ_{LIM1} and θ_{LIM2}.

The logic of FIG. **9***a* performs the operation described in conjunction with FIG. **9***b*, but for simplicity and computational convenience, the processing is performed in terms of parameters $\phi=\cos\theta$, $\phi_{LIM1}=\cos\theta_{LIM1}$, and $\phi_{LIM2}=\cos\theta_{LIM2}$. In FIG. **9***b*, the logic begins at a START block **910** and flows to a block **912**, which represents setting of a flag T_{FLAG} to a value of zero. T_{FLAG} may assume logic values of zero or one, with logic zero indicating that the magnetic torquers were not ON during the previous iteration, and logic one indicating that they were previously ON. The logic flows through blocks **910** and **912** only during initial start-up. The logic flows from block **912** to a block **914**, representing the calculation of ϕ as a quotient of products

$$\phi=Bh_e/|h_e|\cdot|B| \tag{11}$$

From block **914**, the logic flows to a decision block **916**, in which the condition of T_{FLAG} is examined. If T_{FLAG} has a value of one (the torquers were previously ON), the logic exits from decision block **916** by the YES output, and flows to a further decision block **918**. Decision block **918** compares the absolute value of ϕ with ϕ_{LIM1} If $|\phi|$ is greater than ϕ_{LIM1}, the logic exits from decision block **918** by the YES output. This corresponds to a condition in which $\theta \leqq \theta_{LIM1}$ in FIG. **9***b*. From the YES output of decision block **918**, the logic flows to block **920**, representing the setting of $M_{COMMAND}$ to zero, which corresponds to a no-torquing command. The state of T_{FLAG} is set to zero in block **922**, and the logic returns by a path **924** to block **810** of FIG. **8** in preparation for another iteration.

If the torquers were previously ON, as described above, and ϕ is less than, or equal to ϕ_{LIM1} ($\theta > \theta_{LIM1}$ in FIG. **9***b*), the logic leaves decision block **918** of FIG. **9***a* by the NO output, and flows to a block **934**, representing the setting of $M_{COMMAND}$ to equal M_3. This causes the torquers to operate at a magnitude established by M_3, where the magnitude may be accomplished by an appropriate percentage of pulse width modulation. T_{FLAG} is set to a logic one in block **936**, and the logic returns by path **924** to block **810** for another iteration.

If the torquers were previously OFF the logic flow of FIG. **9***a* leaves decision block **916** by the NO output, and flows by a path **926** to a decision block **928**. Decision block **928** compares the absolute value of ϕ with ϕ_{LIM2}. If $|\phi| < \phi_{LIM2}$ (corresponding to $|\theta| > \theta_{LIM2}$ in FIG. **9***b*) the logic leaves decision block **928** by the YES output, and flows by a logic path **930** to block **934**, representing the setting of $M_{COMMAND}$ to equal the value of M_3. This results in torquing of the spacecraft. The logic then flows to block **936**, representing setting of T_{FLAG} to the ONE state. The logic returns by logic path **924** to block **810** to begin another iteration.

If the torquers were previously OFF as mentioned above, the logic reaches decision block **928** of FIG. **9***a*. If $|\phi| \geqq \phi_{LIM2}$ (corresponding to $\theta \leqq \theta_{LIM2}$ in FIG. **9***b*), the logic leaves decision block **928** by the NO output, and proceeds by a logic path **938** to pass in sequence through blocks **920** and **922**, representing no torquing and the setting $T_{FLAG}=0$, respectively, following which another iteration begins.

The dead zone operates in the following manner. If 0 lies between θ_{LIM1} and θ_{LIM2}, θ is greater than θLIM1 and less than θ_{LIM2}, corresponding to $\phi < \phi_{LIM1}$ and $\phi > \phi_{LIM2}$. The logic must flow through either decision block **918** or **928** in order to reach the torquing control condition, represented by blocks **934** and **936**. When the logic flows through decision block **916** and $\phi < \phi_{LIM1}$, the logic always exits by the NO path, thereby flowing to blocks **934** and **936**. When the logic flows through decision block **928**, with $\phi > \phi_{LIM2}$, it always exits by the NO path to blocks **920** and **922**. Since one path represents torquing and the other path represents no torquing, a dead zone exists.

Other embodiments of the invention will be apparent to those skilled in the art. For example, the reaction or momentum wheels may exceed three in number, and may be in arbitrary orientations. The magnetic torquers may have any arbitrary orientation relative to the spacecraft body, and may exceed three in number. Also, the magnetic torquers may have different sizes and/or different magnetic moments. While the satellite orbit has been described as a polar orbit, the advantages of the system may be realized for other orbits.

In addition, the torquing logic of FIG. **9***a* can be implemented by weighting the commanded dipole, M_3, by a factor, Γ, calculated as $\Gamma = 1-|\phi|$ where ϕ is calculated as in block **914**.

What is claimed is:

1. A method for maintaining satellite attitude, comprising the steps of:

measuring the direction of the ambient magnetic field in the vicinity of said satellite;

determining the direction of the net wheel momentum of said satellite;

determining the angle between said direction of said magnetic field and said direction of said momentum;

comparing said angle with a predetermined value; and

energizing magnetic torquers for torquing said satellite for reducing said wheel momentum only when said angle exceeds said predetermined value.

2. A method according to claim 1 further comprising the step of measuring the magnitude of said ambient magnetic field; and

in said step of energizing magnetic torquers, performing the further step of controlling the magnitude of the energization in response to said magnitude of said ambient magnetic field.

3. A method for maintaining satellite attitude, comprising the steps of:

measuring the direction of the ambient magnetic field in the vicinity of said satellite;

determining the absolute magnitude of said ambient magnetic field;

determining the direction of the net wheel momentum of said satellite;

determining the angle between said direction of said magnetic field and said direction of said net wheel momentum;

comparing said angle with a predetermined value;

establishing the magnitude by which magnetic torquers are to be energized in response to said magnitude of said ambient magnetic field, by the further steps of: (a) determining the absolute magnitude of the sum of the spacecraft body, ancillary panel and wheel momenta to generate a momentum error magnitude; (b) comparing said absolute magnitude of said ambient magnetic field with a limiting value of said field; (c) comparing said momentum error magnitude with a limiting momentum magnitude; and (d) multiplying said magnitude of said energization by a multiplier other than unity only when said absolute magnitude of said ambient magnetic field exceeds said limiting value of said field and said momentum error magnitude is less than said limiting momentum value; and

energizing said magnetic torquers for torquing said satellite for reducing said wheel momentum only when said angle exceeds said predetermined value.

4. A method according to claim 3 further comprising the step of:

at least when said absolute value of said ambient magnetic field is less than said limiting value of said field and said momentum error magnitude is greater than said limiting momentum value, multiplying said magnitude of said energization by a value of unity.

5. A method for maintaining satellite attitude, comprising the steps of:

measuring the direction of the ambient magnetic field in the vicinity of said satellite;

determining the direction of the net wheel momentum of said satellite;

determining the angle between said direct of said magnetic field and said direction of said momentum; and

torquing to reduce wheel momentum in proportion to the magnitude of said angle.

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